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RESEARCH MEMORANDUM

FLIGHT MEASUREMENTS OF THE PRESSURE DISTRIBUTION ON THE WING OF THE X-1 AIRPLANE (10-PERCENT-THICK WING) OVER A CHORDWISE STATION NEAR THE MIDSPAN, IN LEVEL FLIGHT AT MACH NUMBERS FROM 0.79 TO 1.00 AND IN A PULL-UP AT A MACH NUMBER OF 0.96

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RESEARCH MEMORANDUM

FLIGHT MEASUREMENTS OF THE PRESSURE DISTRIBUTION ON THE WING OF THE X-1 AIRPLANE (10-PERCENT-THICK WING) OVER A CHORDWISE STATION NEAR THE MIDSPAN, IN LEVEL FLIGHT AT MACH NUMBERS FROM 0.79 TO 1.00 AND IN A PULL-UP AT A MACH NUMBER OF 0.96

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SUMMARY

Measurements of the chordwise pressure distribution over the 10-percent-thick wing of the X-1 research airplane have been made at a section near the midspan of the left wing. Data presented are for a Mach number range from 0.79 to 1.00 at a section normal-force coefficient of about 0.32 and for section normal-force coefficients up to 1.00 at a Mach number of approximately 0.96. The results show that the section center of load moves aft from about 32 percent chord at Mach number 0.79 to 40 percent chord at Mach number 0.84, and then forward to 18 percent chord at Mach number 0.89. The section center of load moves aft to 45 percent chord at Mach number 0.95 and then remains approximately constant at Mach numbers up to 1.00.

At a section normal-force coefficient of 0.32 a shock exists on the upper surface at the lowest test Mach number of 0.79 and supersonic flow exists over approximately 50 percent of the chord on the upper surface. The first indication of a shock on the lower surface occurs at a Mach number of about 0.84. At Mach numbers above 0.95 the shocks on both surfaces occur near the trailing edge and the pressure distribution over both surfaces is quite similar.

An increase in the normal-force coefficient at a Mach number of approximately 0.96 causes a slight increase in the section stability at the higher normal-force coefficients.

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INTRODUCTION

During the course of tests to determine the spanwise and chordwise loading of the 10-percent-thick wing of the X-1 airplane, various stations are being instrumented to measure the pressures over the upper and lower surfaces individually. The data presented herein represent the results of individual surface measurements over a chordwise station at 64.4 percent of the semispan.

SYMBOLS

M	free-stream Mach number
n	normal-load factor
W	airplane weight, pounds
S	wing area, square feet
C_{N_A}	airplane normal-force coefficient (nW/qS)
b	wing span, feet
c	local wing chord parallel to plane of symmetry, feet
x	chordwise location from leading edge, feet
y	spanwise location from airplane center line, feet
q	free-stream dynamic pressure, pounds per square foot
p_o	free-stream static pressure, pounds per square foot
p	local static pressure, pounds per square foot
P	pressure coefficient $\left(\frac{p - p_o}{q}\right)$
P_{cr}	pressure coefficient for sonic velocity
P_U	pressure coefficient on upper surface
P_L	pressure coefficient on lower surface
P_R	resultant pressure coefficient $\left(P_L - P_U\right)$

c_n section normal-force coefficient $\left(\int_0^1 P_R \, d \frac{x}{c}\right)$

$$c_{m_{c/4}} = \int_0^1 - P_R\left(\frac{x}{c} - 0.25\right) d \frac{x}{c}$$

δa_L left aileron angle, (down deflection positive), degrees

DESCRIPTION OF AIRPLANE AND TEST SECTION

The X-1 research airplane used in these tests is shown in figure 1. A three-view drawing of the airplane showing the general over-all dimensions is given as figure 2.

The airplane has a 10-percent-thick wing and incorporates a modified 65-110 airfoil section. At the test section the trailing-edge cusp was replaced by a straight taper from the 85-percent-chord point to the trailing edge (reference 2). The profile and ordinates of the modified section are presented in table I. The wing test section was located at approximately the midspan of the exposed left wing panel and included the inboard end of the aileron (fig. 3). The area about the test section was relatively smooth during the tests but no refined filling or smoothing was attempted.

INSTRUMENTATION

Standard NACA recording instruments were used to obtain airspeed, pressure altitude, normal acceleration, and control positions. Wing-surface pressures were measured by an NACA recording multiple-pressure manometer. All records were synchronized by a common timer.

Free-stream static and dynamic pressures were recorded from an NACA high-speed pitot-static head located at the left wing tip. The static vents were located approximately 0.96 of the local chord ahead of the wing.

Wing-surface pressures were measured from flush-type orifices installed in the wing skin. The orifices were connected to the instrument compartment by $\frac{1}{8}$-inch inside-diameter aluminum tubing. Rubber tubing of $\frac{3}{16}$-inch inside diameter was used between the aluminum tubing and the manometer cells. The average length of aluminum tubing was approximately 8 feet. About three feet of rubber tubing was used on each line.

Normal acceleration was measured near the center of gravity of the airplane.

ACCURACY

The accuracy of the test results is estimated to be within the following limits:

Mach number	±0.01
p	±0.02q
c_n	±0.05
$c_{m_{c/4}}$	±0.006

METHODS

The pressure cells of the multiple manometer were vented to the fuselage instrument compartment; thus, pressures on the wing surface were measured relative to the existing compartment static pressure. Static pressure at the pitot-static head was also measured relative to compartment pressure. The measured static pressure at the boom was corrected to free-stream static pressure by the use of the radar tracking method of reference 1. Ground tests were made to determine any effects of lag that might be present in measuring the wing-surface pressures. These tests show that the effects of lag were negligible and have been neglected in these data.

TESTS

The data presented herein were obtained during a three-rocket flight at 40,000 feet altitude. A pull-up was made at the end of the level run by using both the elevator and adjustable stabilizer. Continuous records of pressure distribution were obtained during the level run ($M = 0.79$ to 1.00) and during the pull-up to a load factor of 3.2 at a Mach number of approximately 0.96.

PRESENTATION OF DATA

Presented in figure 4 are pressure distributions for the upper and lower surfaces of the wing test section for a Mach number range from 0.79 to 1.00. These distributions have been selected for a section normal-force coefficient of about 0.32. Pressures were not obtained at all orifices on the section because of leaks in some pressure lines within the wing. Included in the figures are values of free-stream Mach number, airplane normal-force coefficient, integrated section normal-force coefficient, left-aileron deflection, and pressure coefficient for sonic velocity, for each distribution.

Pressure distributions obtained during the pull-up at a Mach number of approximately 0.96 are presented in figure 5. The data cover a range of section normal-force coefficients from 0.34 to 1.00. Values of free-stream Mach number, airplane normal-force coefficient, integrated section normal-force coefficient, left-aileron deflection and pressure coefficient for sonic velocity are included for each pressure distribution.

The total-load distributions over the section derived from the faired distributions of individual surfaces of figure 4 are presented as figure 6. The data cover a Mach number range from 0.79 to 1.00 for a section normal-force coefficient of about 0.32. Values of free-stream Mach number, airplane normal-force coefficient, integrated section normal-force coefficient, and left-aileron deflection are included for each distribution.

Figure 7 presents the total-load distributions over the section derived from the faired distributions of figure 5. The data cover a range of section normal-force coefficients from 0.34 to 1.00 at a Mach number of approximately 0.96. Values of free-stream Mach number, airplane normal-force coefficient, integrated section normal-force coefficient, and left aileron deflection are included for each distribution.

A figure showing the wing section characteristics for the Mach number range investigated is shown as figure 8. The section pitching-moment coefficient has been derived from the center-of-pressure curve for a section normal-force coefficient of 0.32. Also shown on the plot are data obtained with the 8-percent-thick wing X-1 airplane as given in reference 3.

Figure 9 presents the variation of the section center of pressure and the section pitching moment with section normal-force coefficient for values of section normal-force coefficient from 0.34 to 1.00 at a Mach number of approximately 0.96.

DISCUSSION

Upper-Surface Pressure Distribution

For the given section normal-force coefficient, at the lowest Mach number of these data (M = 0.79), the pressure coefficient for sonic velocity has been exceeded and supersonic flow exists over approximately 50 percent of the chord on the upper surface (fig. 4). The presence of a compression shock is indicated by an abrupt pressure recovery at about the 55-percent-chord point.

Figure 4 shows that, for a given section normal-force coefficient, there is a gradual transition of load distribution over the chord as the Mach number increases from 0.79 to 1.00. The compression shock on the upper surface moves toward the trailing edge as the Mach number increases. It may be seen that the transition results in a shift of the load center to the rear with increasing Mach number. As will be shown in the following discussion of selected Mach number ranges, the load-center shift is a result of the shock movement toward the rear and the extension of the region of supersonic flow expansion over the airfoil.

Mach numbers 0.79 to 0.89.- At Mach numbers from 0.79 to 0.89, for the given section normal-force coefficient, the shock remains at about 50 to 60 percent chord. The pressures forward of the shock remain approximately constant. The pressures behind the shock, however, change considerably. The gradual pressure recovery behind the shock at a Mach number of 0.79 indicates that the flow has not separated from the surface. As the Mach number increases to 0.89 the pressures behind the shock increase, negatively, to a nearly constant value and indicate that the flow has separated from the surface of the airfoil.

Mach numbers 0.89 to 0.95.- Increasing the Mach number from 0.89 to 0.95 causes the shock to move to a location near the trailing edge with a corresponding increase in trailing-edge loads. Little change occurs in the shape of the pressure distribution forward of the 55-percent-chord point, but a progressive reduction in load occurs.

Mach numbers 0.95 to 1.00.- As the Mach number is increased from 0.95 to 1.00, the shape of the distribution over the upper surface remains approximately the same, but the magnitudes of the loads over the upper surface are reduced slightly.

An increase in the normal-force coefficient at a Mach number of 0.96 (fig. 5) causes the pressure distribution to change from one with maximum negative pressure coefficient near the trailing edge to a distribution with approximately constant pressure coefficient over the chord.

Lower-Surface Pressure Distribution

As shown in figure 4 a similar change in load distribution, due to changes in Mach number, occurs over the lower surface as occurs over the upper surface. With the formation of shock and increase in Mach number, the load center shifts toward the rear. For the given normal-force coefficient the shock on the lower surface forms rearward of the upper surface shock, at a higher Mach number and moves to the rear more rapidly.

Mach numbers 0.79 to 0.89.- At Mach numbers below 0.84 there is no indication of a shock on the lower surface at the given normal-force coefficient, and the shape of the distribution over the lower surface remains essentially unchanged. At a Mach number close to 0.84 a shock forms on the lower surface at about 60 percent of the chord. As the Mach number increases, the shock moves steadily rearward and at a Mach number of 0.89 occurs at 80 to 85 percent of the chord.

Mach numbers 0.89 to 0.95.- At the given normal-force coefficient, as the Mach number increases from 0.89 to 0.95 the shock on the lower surface moves to a position near the trailing edge. The distribution over the forward 55 percent of the chord remains essentially unchanged.

Mach numbers 0.95 to 1.00.- An increase in Mach number from 0.95 to 1.00 causes very little change in the shape of the pressure distribution over the lower surface.

As the normal-force coefficient is increased at a Mach number of 0.96 the pressure coefficients over the lower surface tend to become more positive so that a rearward shift of the lower-surface center of pressure occurs (fig. 5).

Section Loads

At a Mach number of 0.79, for the given normal-force coefficient, most of the load is carried by the forward 55 percent of the chord. At Mach numbers above 0.95 the load distribution is approximately rectangular with the load distributed over the entire chord (fig. 6). Between these limits the variation of the shock position on one surface with respect to the other and the different rates of shock movement on each surface causes a region of down load to form aft of the 55-percent-chord point at a Mach number of 0.84. Further increase in the Mach number causes the region of down load to increase in magnitude up to a Mach number of 0.89 and then to decrease in magnitude and disappear at Mach numbers above 0.93.

The changes in load distribution due to Mach number result in considerable changes in section center of pressure and section pitching-moment coefficient. The section center of pressure at a Mach number of 0.79 is located at approximately 32 percent of the chord (fig. 8) and moves rearward to about 40 percent chord as the Mach number increases to 0.84 due to the changes in load over the aft portion of the upper surface. As the Mach number increases to 0.89 the center of pressure moves forward to 18 percent chord, primarily because of the formation and rearward movement of the lower-surface shock. At Mach numbers between 0.89 and 0.95, the center of pressure moves rearward to about 45 percent of the chord because of the rapid rearward movement of the upper-surface shock. In the Mach number range from 0.95 to 1.00, where the shock waves on both the upper and lower surfaces have reached the vicinity of the trailing edge, the center of pressure remains stationary at 45 percent of the chord.

A comparison of the section characteristics for the 8-percent-thick wing (reference 3) and the 10-percent-thick wings shows similar variations over the Mach number range investigated (fig. 8); however, the magnitude of the changes is slightly different. The difference in magnitude of the changes may result from the difference in thickness of the section or the difference in spanwise location of the sections tested. The section used in reference 3 was at 49.1 percent of the semispan whereas the section used herein was at 64.4 percent semispan.

An increase in the normal-force coefficient at a Mach number of approximately 0.96 (fig. 9) causes the section center of pressure to move forward slightly to about 41 percent of the chord at a normal-force coefficient of about 0.7 (fig. 9). At higher normal-force coefficients the center of pressure moves rearward slightly as a result of changes in the lower-surface loads. This change results in a small increase in the stability of the section as shown by the increase in slope of the curve of pitching-moment against normal-force coefficient in figure 9.

CONCLUSIONS

Results of the chordwise-pressure-distribution measurements over the midspan station on the left wing of the X-1 airplane show that:

1. At a section normal-force coefficient of 0.32, in the Mach number region between 0.79 and 1.00, large changes in chordwise center of pressure occur as a result of the formation and movement of shock over the upper and lower surfaces. As the Mach number M increases from 0.79 to 0.84 the center of pressure moves rearward from 32 to 40 percent chord, from $M = 0.84$ to 0.89 the center of pressure moves forward to 18 percent chord, and from $M = 0.89$ to 0.95 the center of pressure moves aft to 45 percent chord. At Mach numbers from 0.95 to 1.00 the center of pressure remains approximately constant.

2. An increase in the normal-force coefficient at a Mach number of approximately 0.96 causes a slight increase in the section stability at the higher normal-force coefficients.

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REFERENCES

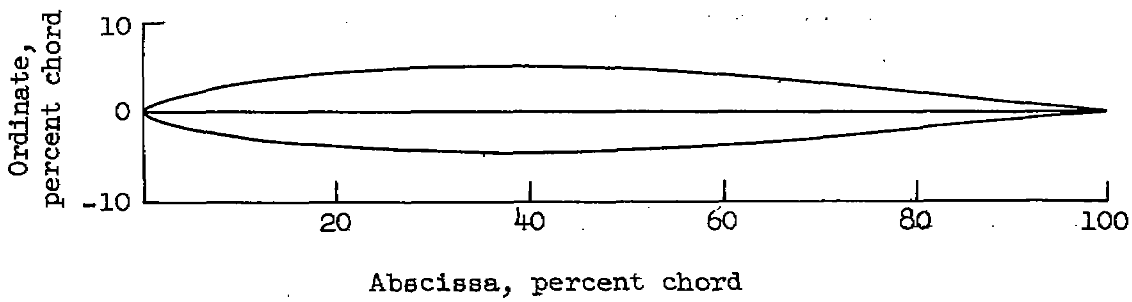
1. Zalovcik, John A., and Wood, Clotaire: Static-Pressure Error of an Airspeed Installation on an Airplane in High-Speed Dives and Pull-Outs. NACA RB L5K29a, 1946.

2. Ormsby, C. A.: Aerodynamic Design of the MX-653 Wing. Rep. No. 44-943-008, Bell Aircraft Corp., June 5, 1945.

3. Beeler, De E., McLaughlin, Milton D., and Clift, Dorothy C.: Measurements of the Chordwise Pressure Distributions over the Wing of the XS-1 Research Airplane in Flight. NACA RM L8G21, 1948.

TABLE I.- PROFILE AND ORDINATES OF THE AIRFOIL SECTION

[Abscissas and ordinates in percent of local chord]



Modified NACA 65-110 airfoil section		
Abscissa	Ordinate	
	Upper surface	Lower surface
0	0	0
.50	.796	-.746
.75	.966	-.896
1.25	1.222	-1.115
2.50	1.667	-1.481
5.00	2.334	-2.018
7.50	2.859	-2.435
10.00	3.298	-2.781
15.00	4.002	-3.329
20.00	4.541	-3.745
25.00	4.951	-4.056
30.00	5.246	-4.274
35.00	5.439	-4.409
40.00	5.532	-4.461
45.00	5.511	-4.416
50.00	5.364	-4.261
55.00	5.078	-3.983
60.00	4.682	-3.611
65.00	4.197	-3.167
70.00	3.642	-2.670
75.00	3.032	-2.137
80.00	2.385	-1.589
85.00	1.721	-1.048
90.00	1.148	-.698
95.00	.574	-.349
100.00	0	0
L.E. radius = 0.687 percent chord		

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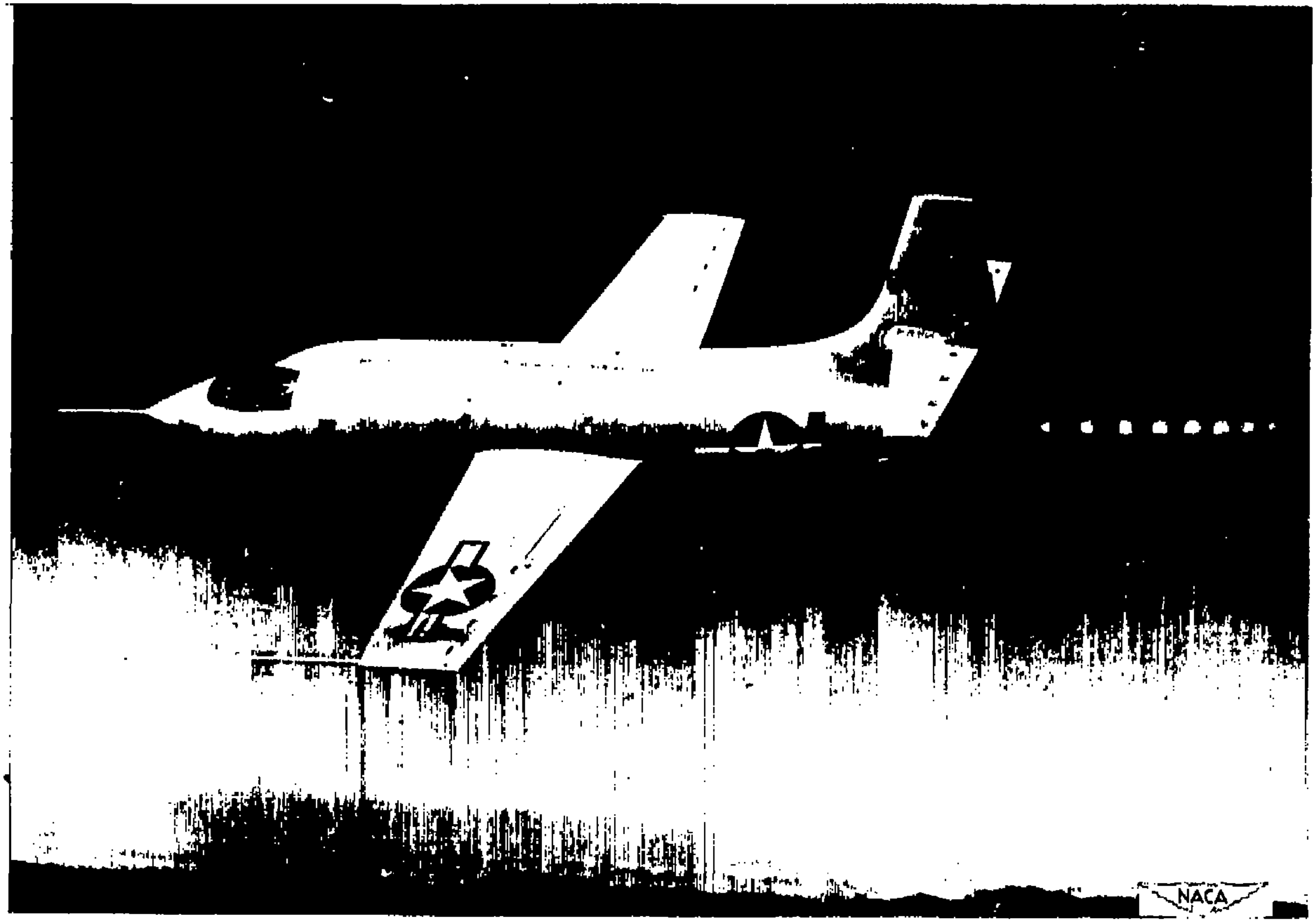


Figure 1.- X-1 airplane in powered flight.

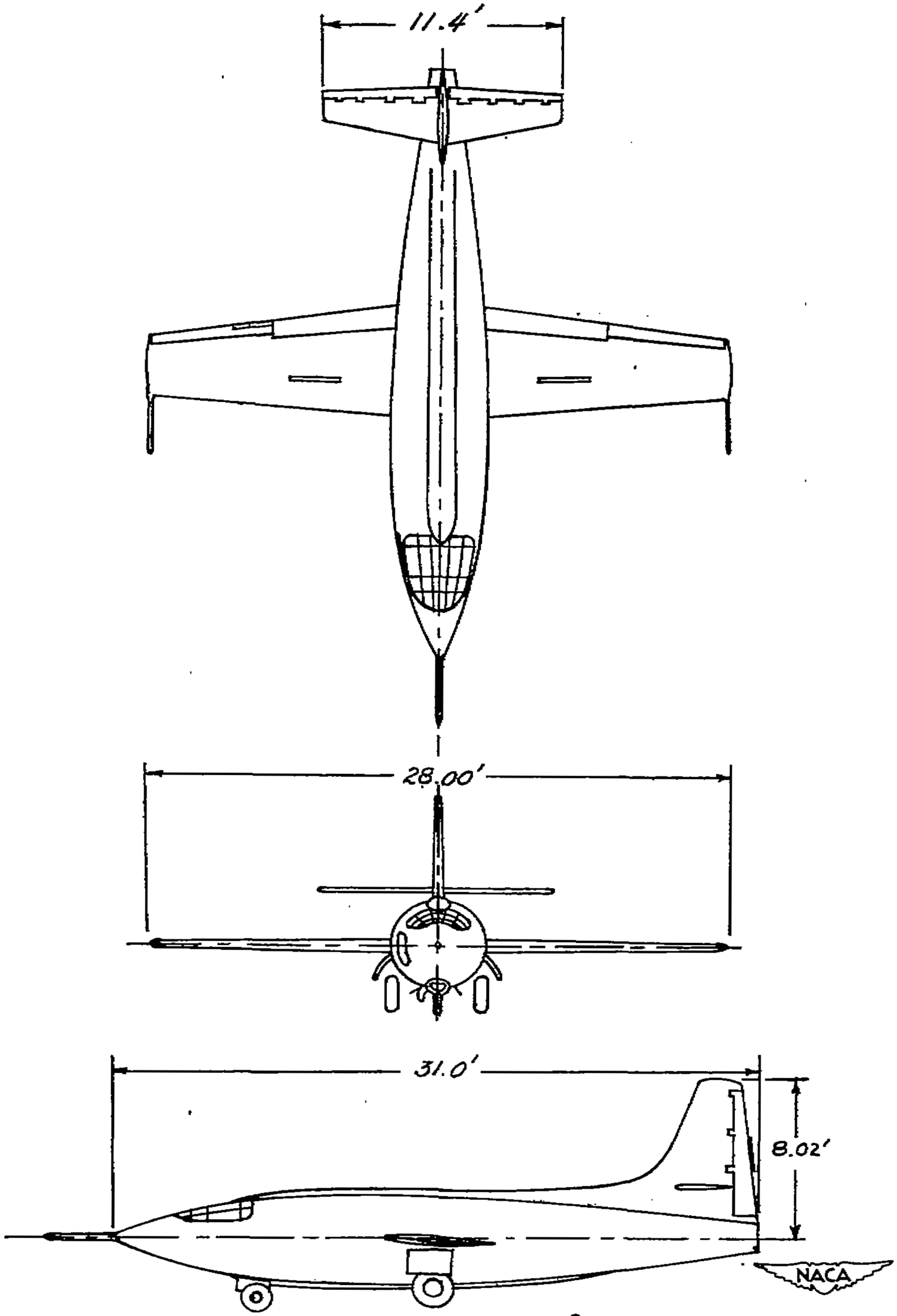
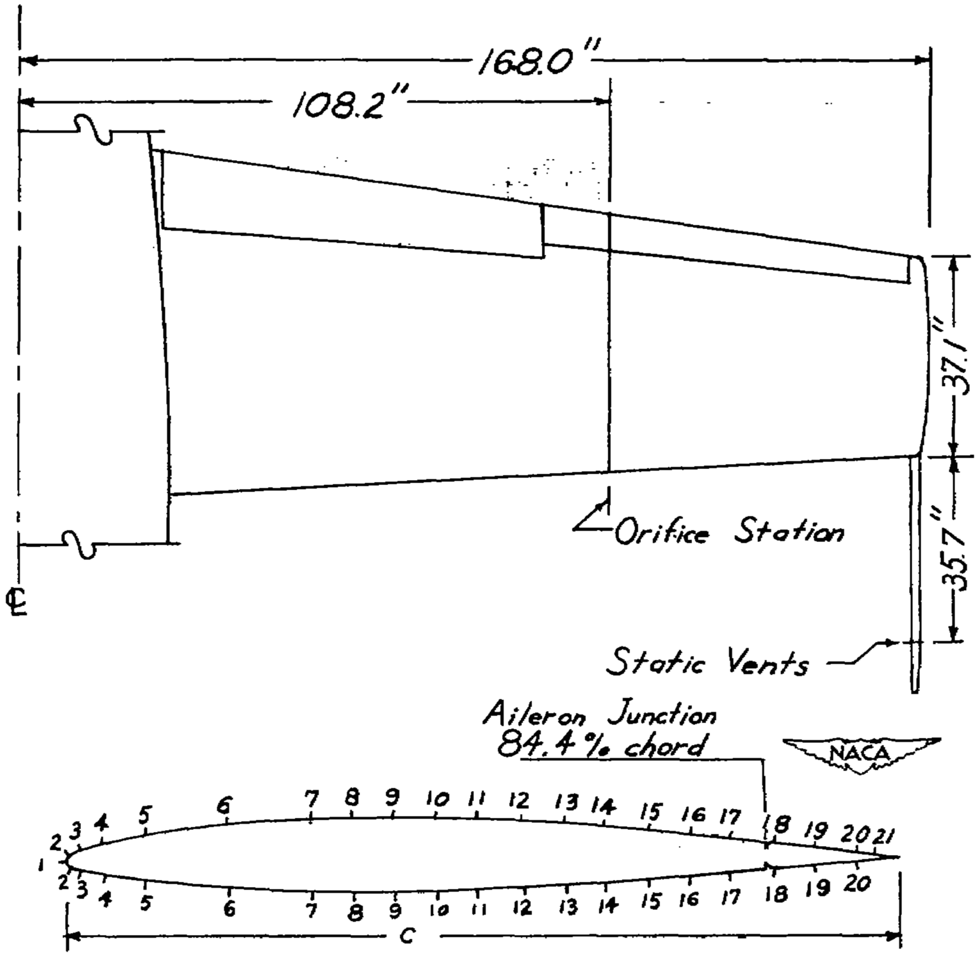


Figure 2.- Three-view drawing of X-1 airplane.



Orifice No.		1	2	3	4	5	6	7	8	9	10
Orifice Location	Upper	0	1.29	2.66	5.16	10.95	19.76	30.00	34.80	40.00	45.15
Percent chord, c	Lower	0	1.38	2.66	5.16	10.95	20.10	30.00	35.10	40.15	45.35

11	12	13	14	15	16	17	18	19	20	21
50.18	55.28	60.80	65.40	69.85	74.40	79.50	85.62	90.00	95.00	97.10
50.30	55.28	60.60	65.60	69.95	74.20	79.70	85.40	90.00	95.00	——

Figure 3.- Location of pressure-measuring orifices for obtaining section pressure distribution.

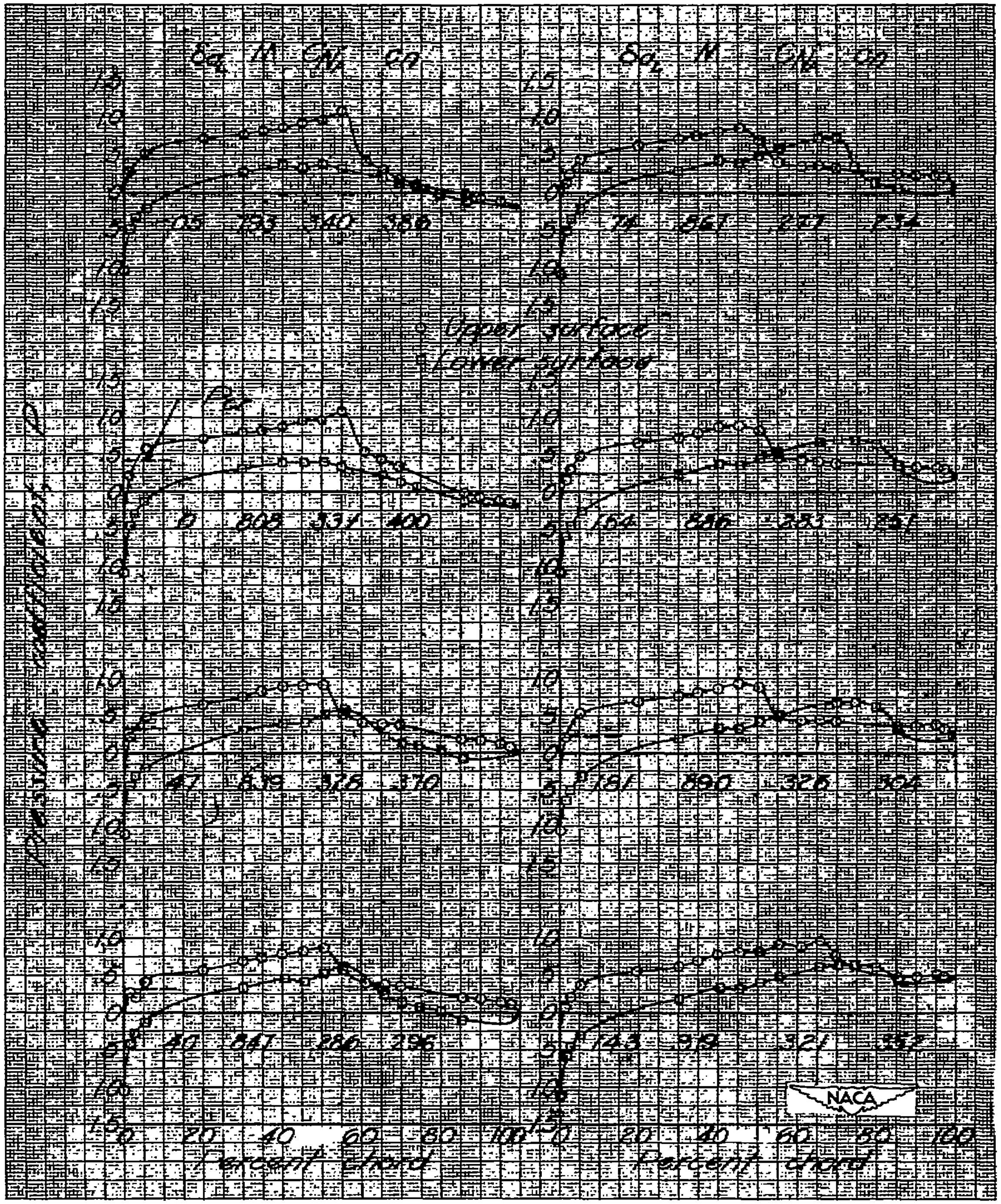


Figure 4.- Pressure distribution over one chordwise wing station of the X-1 airplane for several values of Mach number. $\frac{2y}{b} = 0.644$; $c_n = 0.32 \pm 0.08$; wing thickness, 0.10 chord.

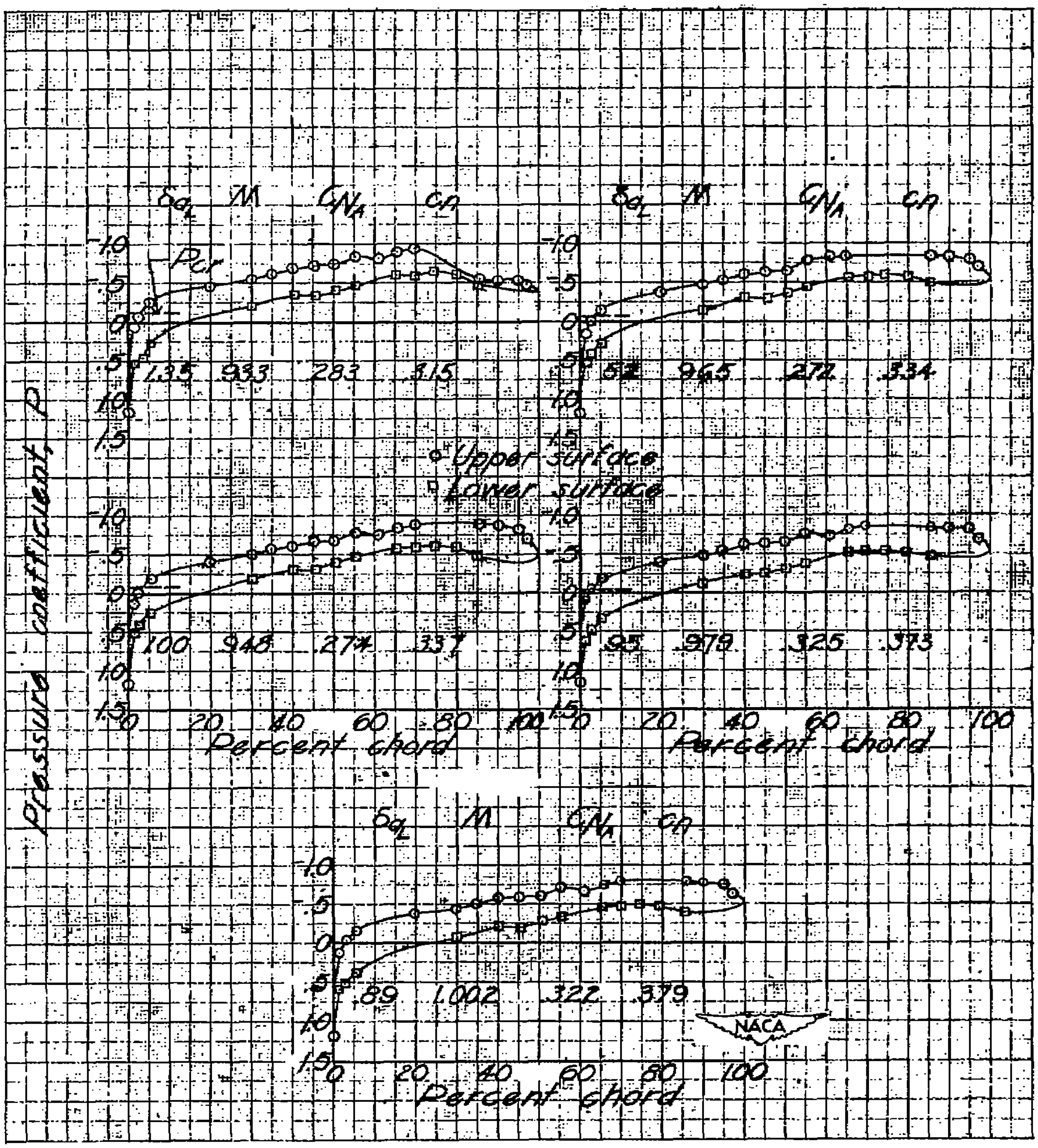


Figure 4.- Concluded.

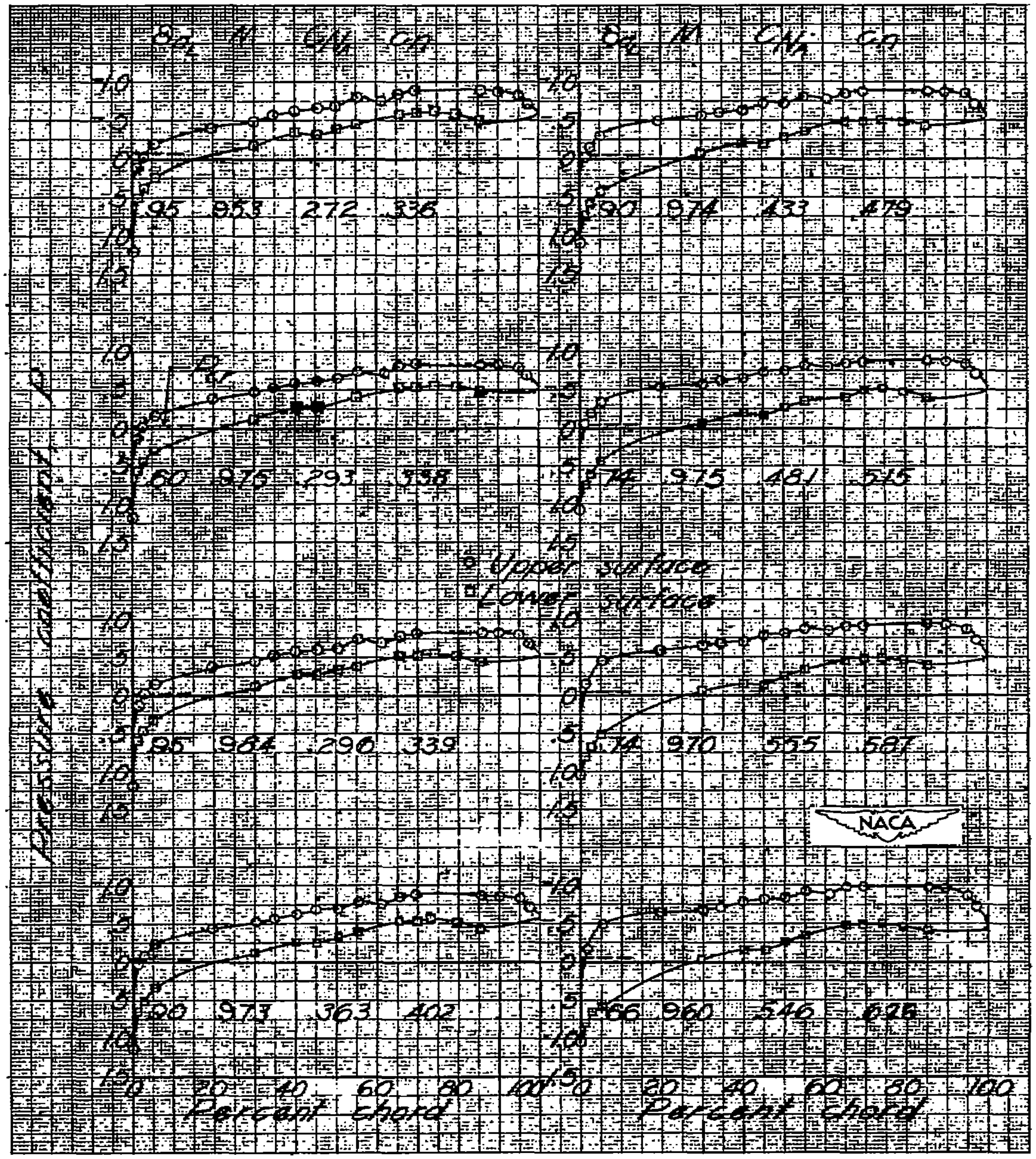


Figure 5.- Pressure distribution over one chordwise wing station of the X-1 airplane for several values of normal-force coefficient. $\frac{2y}{b} = 0.644$; $M = 0.96 \pm 0.02$; wing thickness, 0.10 chord.

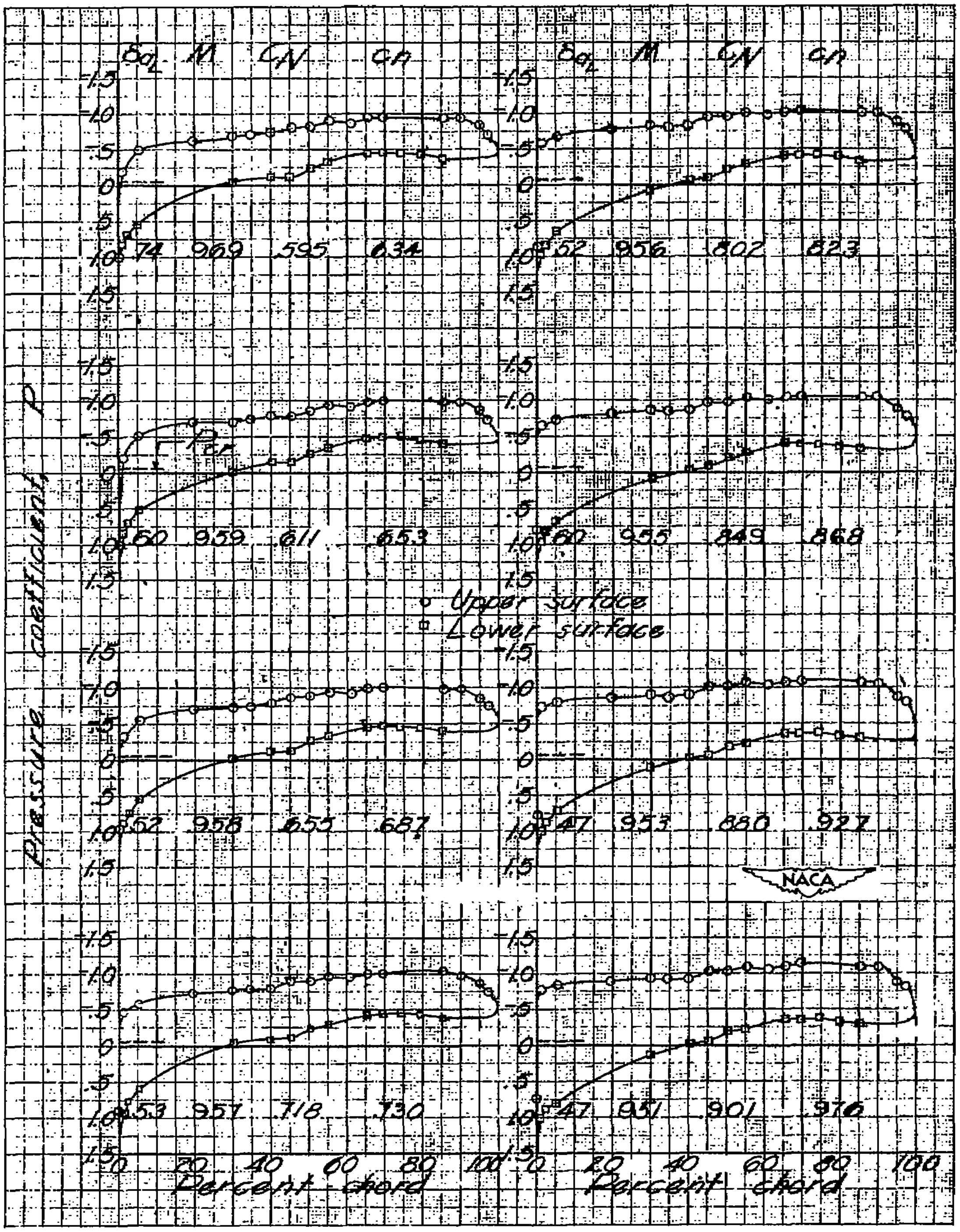


Figure 5.- Continued.

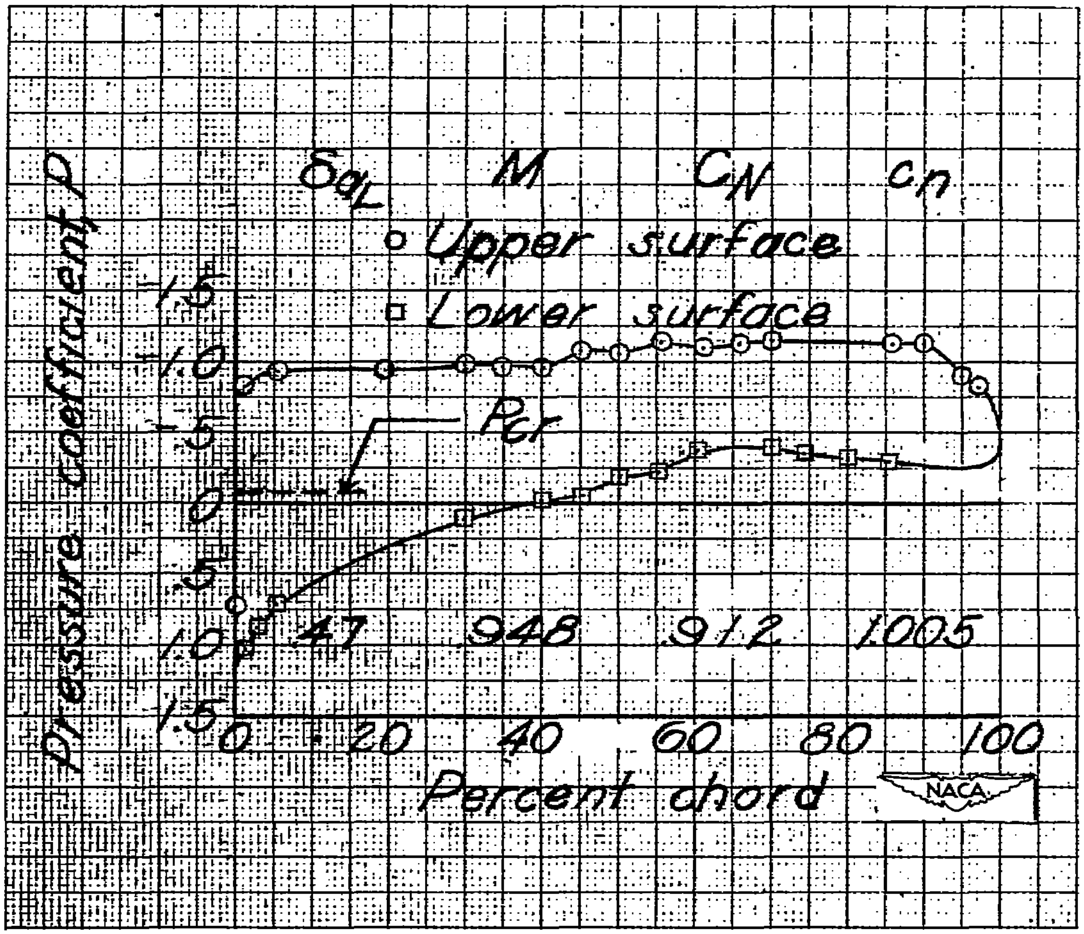


Figure 5.- Concluded.

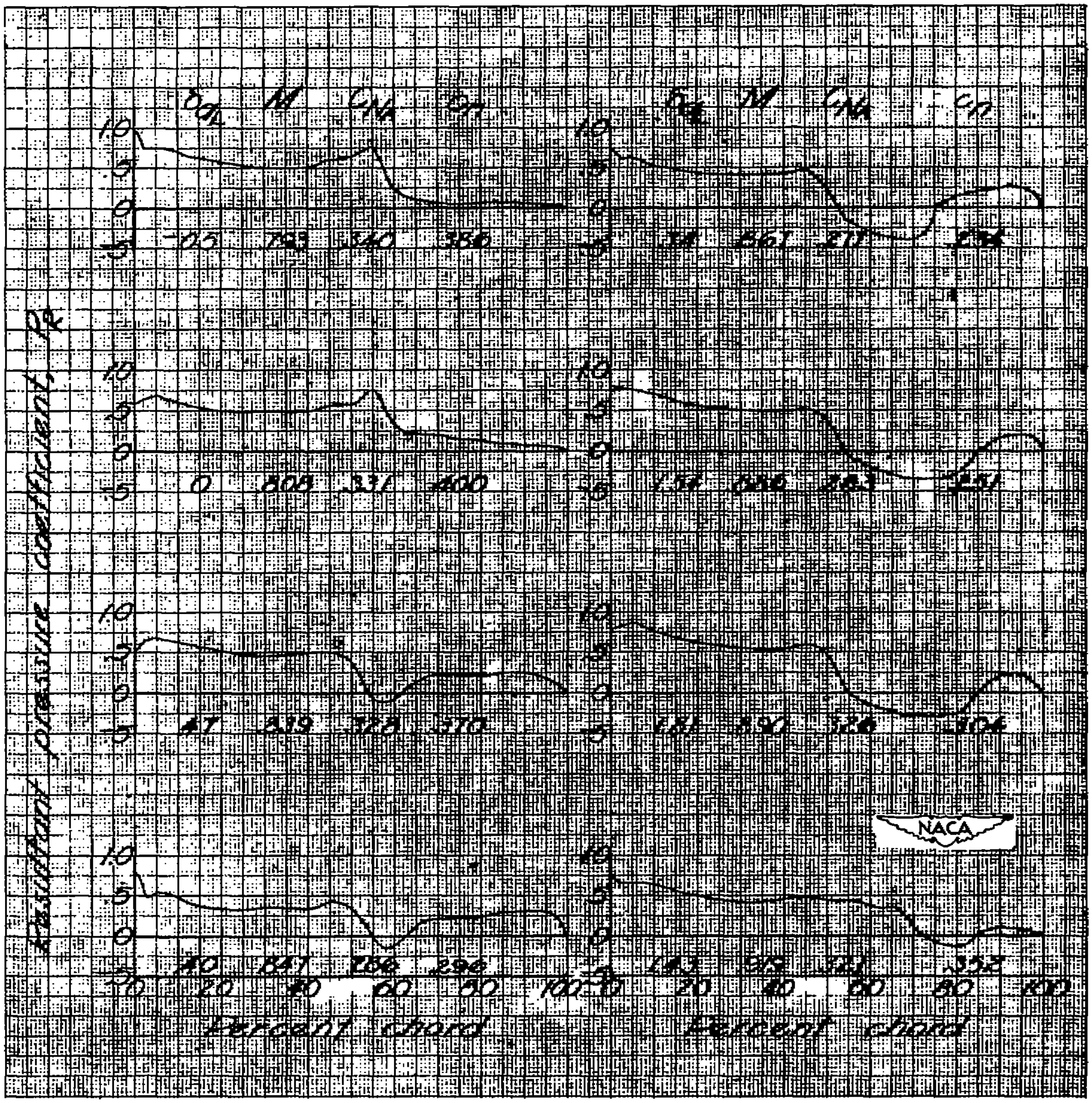


Figure 6.- Chordwise loading over one wing station of the X-1 airplane for several values of Mach number. $\frac{2y}{b} = 0.644$; $c_n = 0.32 \pm 0.08$; wing thickness, 0.10 chord.

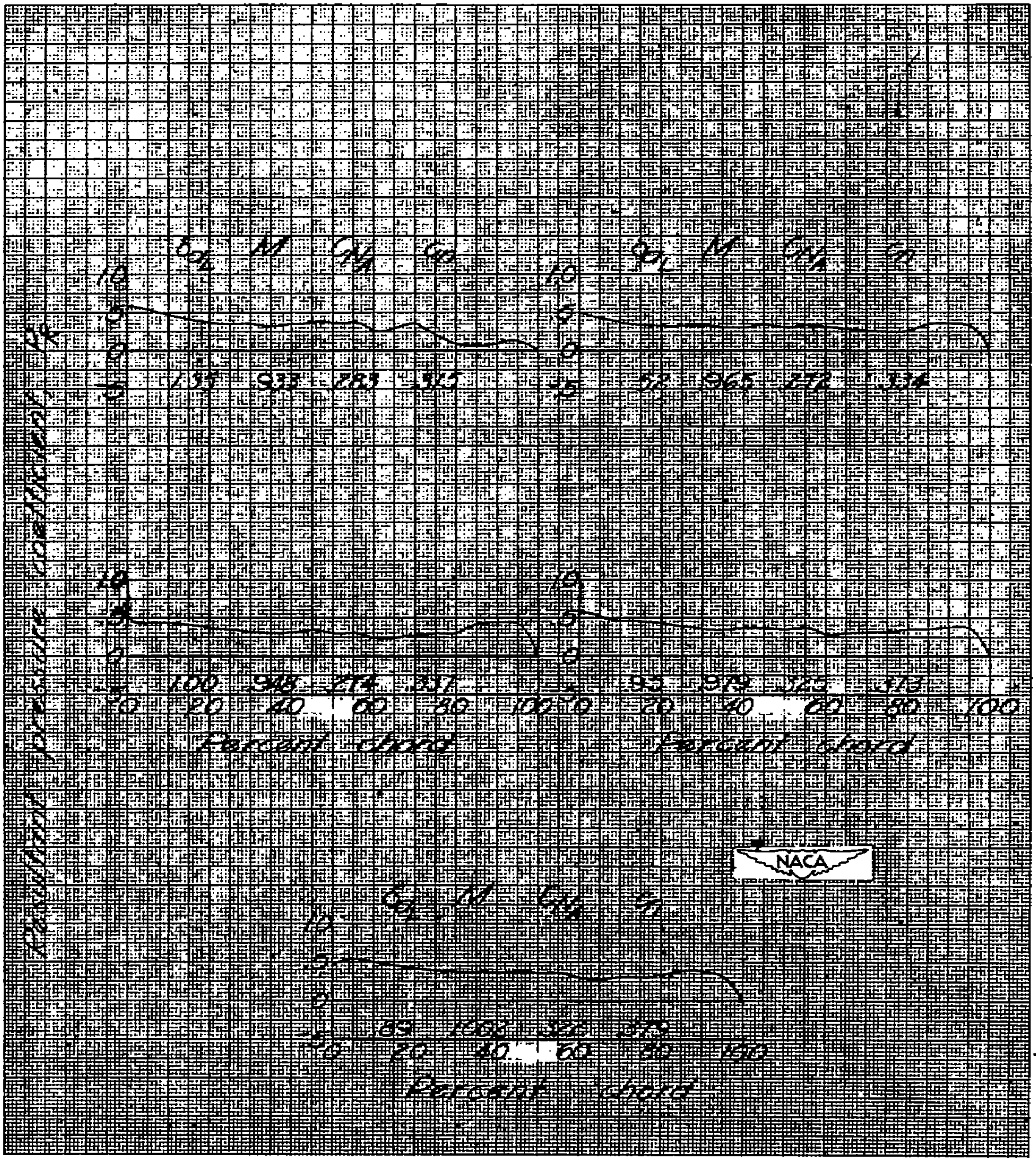


Figure 6.- Concluded.

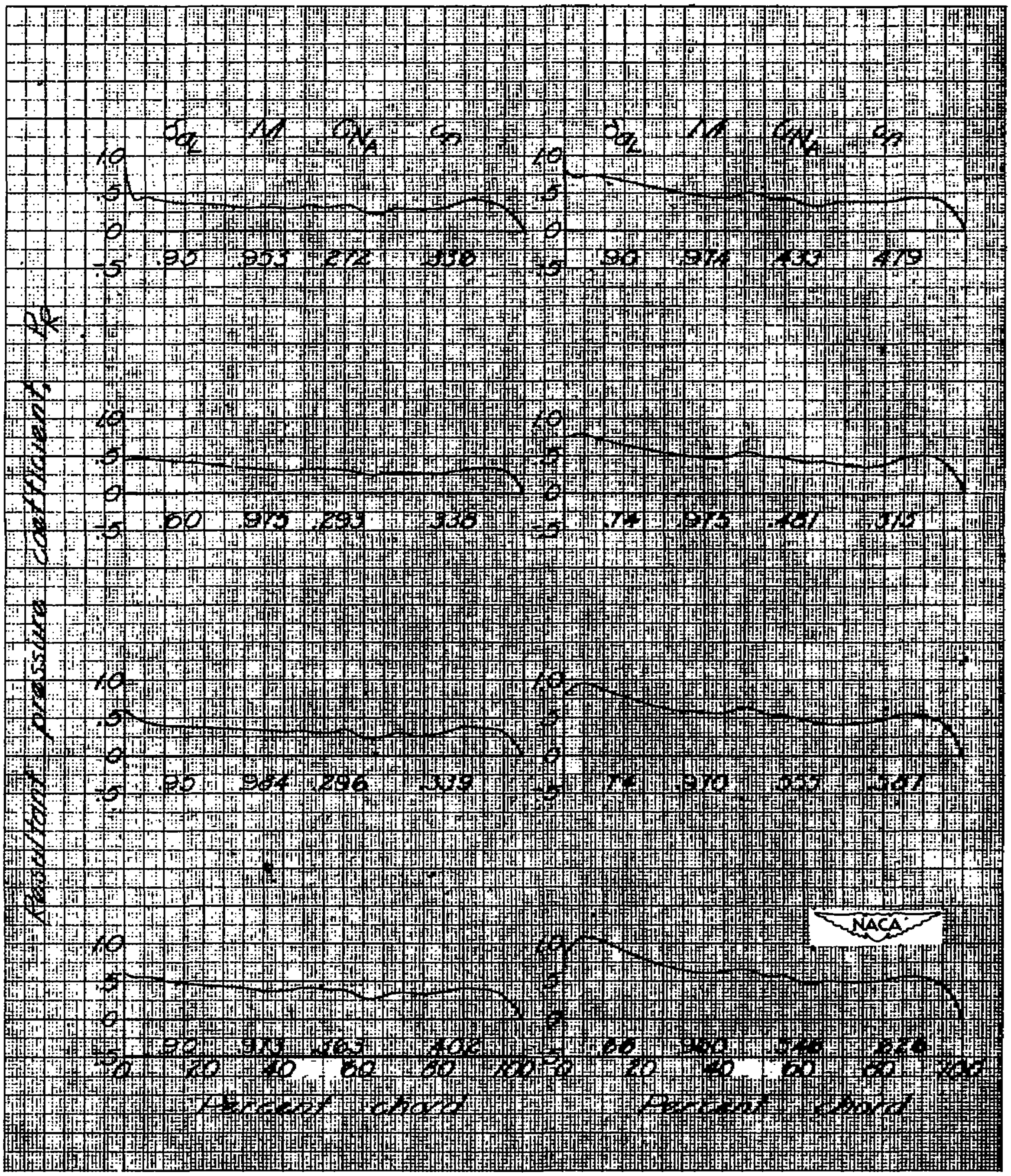


Figure 7.- Chordwise loading over one wing station of the X-1 airplane for several values of normal-force coefficient. $\frac{2y}{b} = 0.644$; $M = 0.96 \pm 0.02$; wing thickness, 0.10 chord.

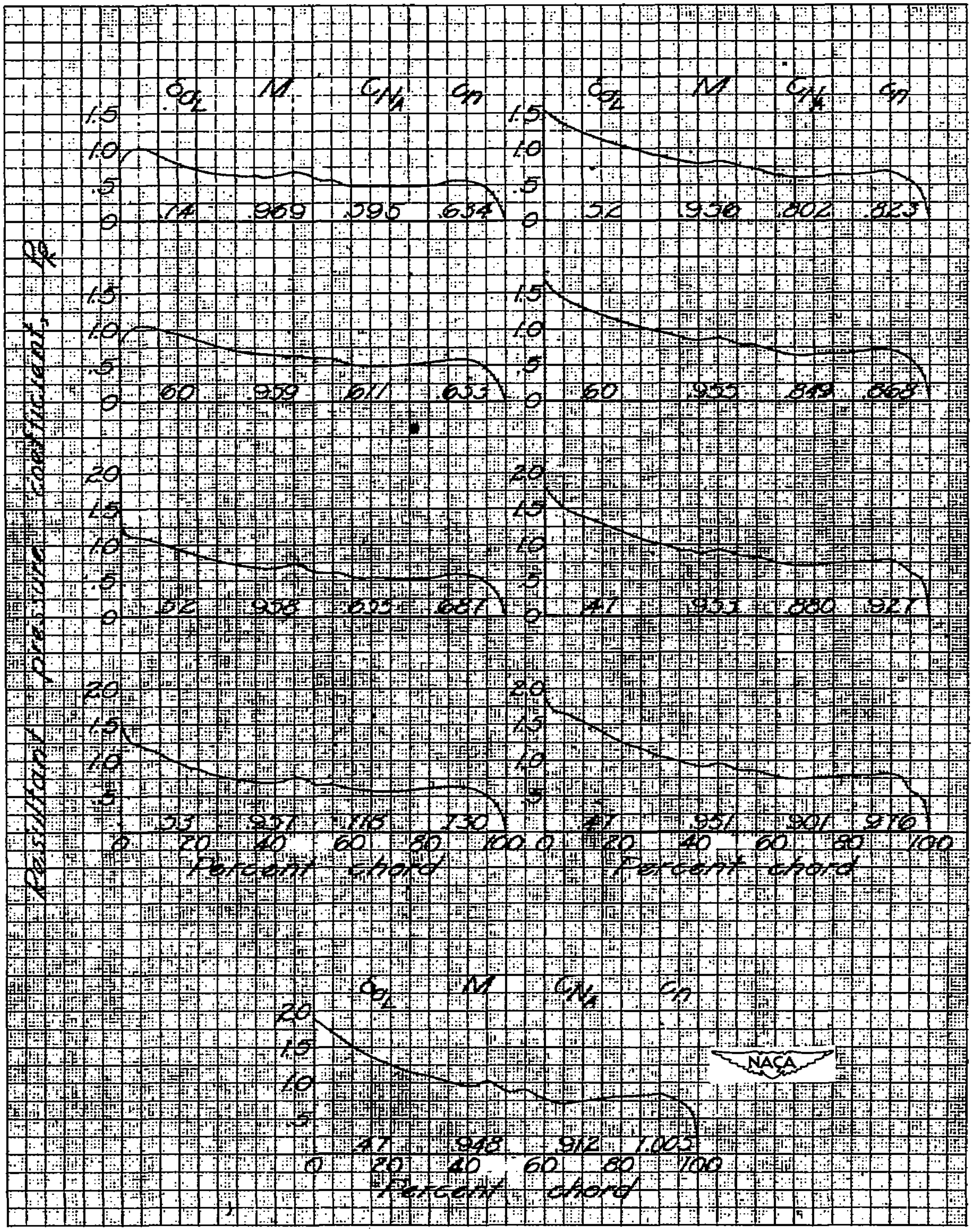


Figure 7.- Concluded.

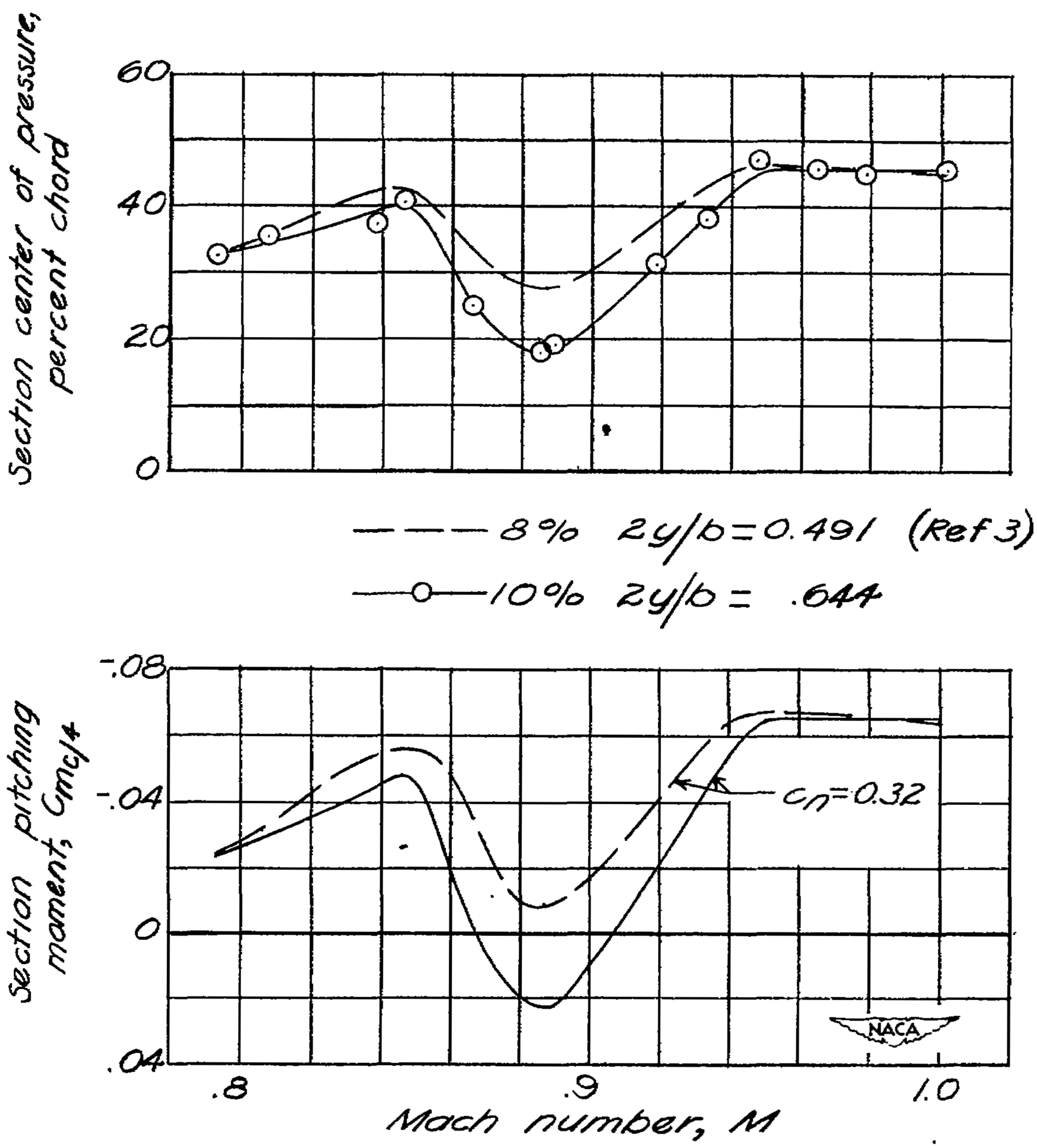


Figure 8.- Variation of section center of pressure and section pitching moment with Mach number. $c_n = 0.32 \pm 0.08$.

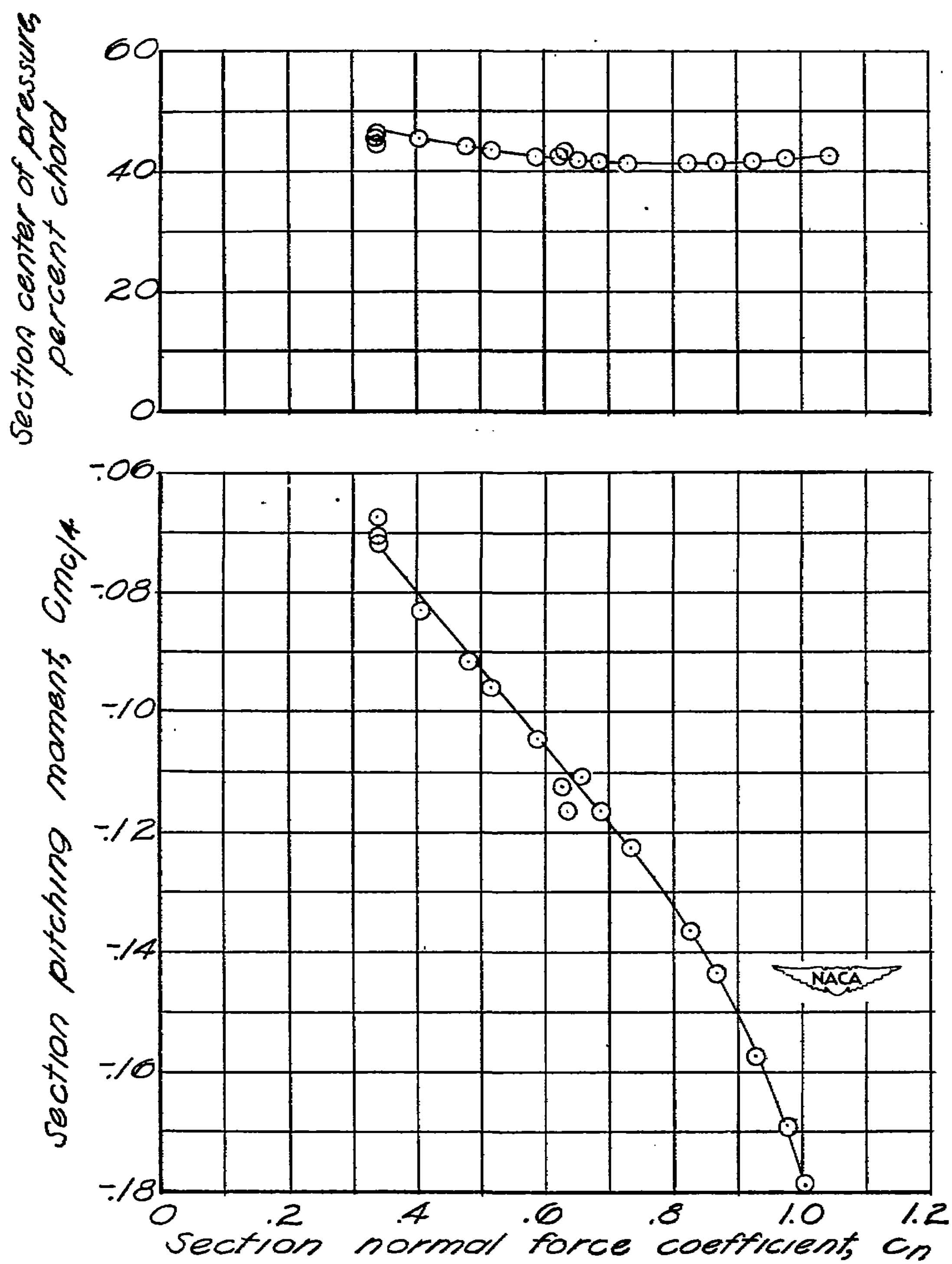


Figure 9.- Variation of section center of pressure and section pitching moment with section normal-force coefficient. M = 0.96 ± 0.02.